

NASA
SPACE VEHICLE
DESIGN CRITERIA
(GUIDANCE AND CONTROL)

NASA SP-8047

SPACECRAFT SUN SENSORS



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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

FOREWORD

NASA experience has indicated a need for uniform design criteria for space vehicles. Accordingly, criteria are being developed in the following areas of technology:

Environment
Structures
Guidance and Control
Chemical Propulsion

Individual components of this work will be issued as separate monographs as soon as they are completed. This document, "Spacecraft Sun Sensors," is one such monograph. A list of all monographs in this series issued prior to this one can be found on the last page of this document.

These monographs are to be regarded as guides to design and not as NASA requirements, except as may be specified in formal project specifications. It is expected, however, that the criteria sections of these documents, revised as experience may indicate to be desirable, eventually will be uniformly applied to the design of NASA space vehicles.

This monograph was prepared under the cognizance of the NASA Electronics Research Center. J. M. Hall and Mark Harris of Exotech, Inc., were committee chairman and program manager, respectively.

The effort was guided by an advisory panel consisting of the following individuals:

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Contributions in the area of design and development practices were also provided by many other engineers of NASA and the aerospace community.

Comments concerning the technical content of these monographs will be welcomed by the National Aeronautics and Space Administration, Office of Advanced Research and Technology (Code RVA), Washington, D.C. 20546.

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SPACECRAFT SUN SENSORS

1. INTRODUCTION

Solar radiation sensing devices have been developed and utilized on a large number of spacecraft for attitude determination, attitude control, and the generation of switching and timing signals. Typically, the combined output of a number of individual sun sensor units (a sun sensor system) is needed to develop the required output information.

The design of a sun sensor system for any spacecraft application must provide acceptable physical characteristics, meet performance specifications, and cope with interference sources and environmental conditions while minimizing penalties to spacecraft design and performance. Physical characteristics include size, weight, and power requirements. The major performance parameters are accuracy, stability, field of view, transfer function characteristics, resolution, and reliability.¹ Interference sources include sunlight reflected from Earth and other planetary bodies, reflection from spacecraft surfaces, and spacecraft electromagnetic fields. Environmental factors to be considered are handling shock, launch vibration, temperature, pressure, contamination, high-energy radiation, and micrometeoroid fluxes.

Other considerations that often influence sensor design are the characteristics of the attitude control system, the spacecraft structure, and the interface constraints imposed by mission objectives and scientific instrumentation. Inadequate attention to design details of this unit can lead to degraded control system performance; excessive power and fuel consumption; program delays; and, in the extreme, failure to achieve mission objectives.

The scope of this document is limited to sensor systems that relate the line of sight to the geometrical center of the Sun to a reference coordinate system associated with the spacecraft. While much of the information contained herein may be applicable to other special purpose sun sensors, such as scanners, limb pointers, feature trackers, eclipse sensors, or guide telescopes, such units are not covered specifically. The electronics associated with the sensor are considered only as interfaces.

¹See appendix for definitions of commonly used performance parameters.

2. STATE OF THE ART

Sun sensors have been used on numerous space vehicles and have, in most cases, performed satisfactorily and reliably. Present development and manufacturing techniques are adequate for most specifications, but testing techniques are generally behind the state of the art of hardware development. A persistent problem is that handling and processing of sensors and related components often expose the units to environments that are more severe than the specified qualification levels. This is particularly true for the thermal, contamination, and shock environments. Difficulties experienced with sun sensors, resulting in costly delays, are often traceable to inadequate test programs or facilities. The need for recognition of potential problems and for inclusion of testability in sensor design is apparent from an examination of past development programs.

2.1 Historical Background

Preceding aerospace applications on rocket flights in the early 1950's, sun sensors were employed for the guidance of ground-based solar telescopes (ref. 1), in automatic sextants (ref. 2), and for the guidance of heliostats on solar furnaces (ref. 3). In the transition from terrestrial to flight applications, the first requirements were miniaturization and ruggedness. Initially, miniature phototubes were shock mounted to survive the launch vibration. Solid-state photoresistive detectors (cadmium sulfide, cadmium selenide) represented a major improvement and were used on later rockets and on some early spacecraft. On the first rocket flight applications, sun sensors were used to determine the attitude of the vehicle. Later, they furnished control signals for the orientation of scientific instruments with respect to the Sun (ref. 4).

The development of solar cells for spacecraft power supplies provided indirect benefits to sun sensor instrument technology. Silicon solar cells suitably modified, photodiodes, and light-activated semiconductor controlled rectifiers (photo-SCR's) were rapidly accepted and incorporated into sun sensor design. Today, the vast majority of spacecraft sun sensors employ these devices as detectors.

Sun sensor systems have been effectively employed on many spacecraft to perform such functions as orientation of scientific instruments, attitude determination, spin-rate detection, switching, and timing (refs. 5, 6, and 7). However, the increasing sophistication of experiments and missions and the stringent environmental and reliability requirements necessary for advanced planetary missions provide impetus for continued improvement of sun sensor design.

2.2 Sensor Description and Model

Sensors that fall within the scope of this document indicate the orientation of the Sun with respect to a reference coordinate system by detecting the intensity difference between

radiation arriving from the solid angle determined by the Sun's boundaries and that arriving from adjacent regions within the sensor's field of view. Although hardware designs vary widely, sun sensor systems are designed to provide two basic output signal formats, analog and digital. In the analog systems the output is a continuous function of the angle of incidence; in the digital systems the output is discrete. Figures 1 and 2 illustrate the principles of each type.

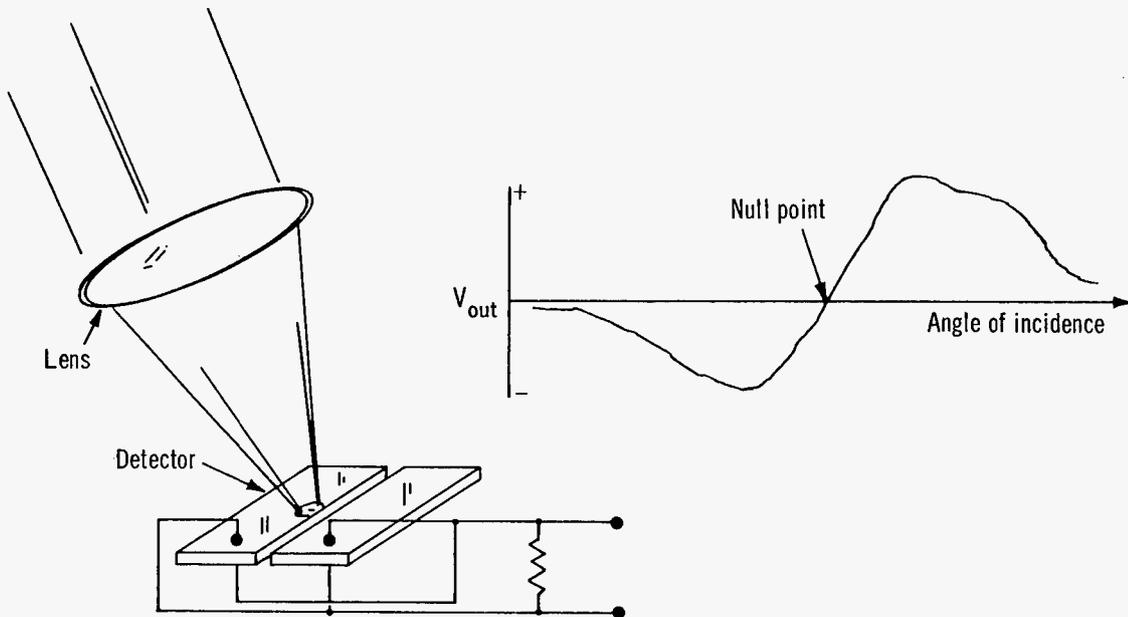


Figure 1.—Analog sun sensor.

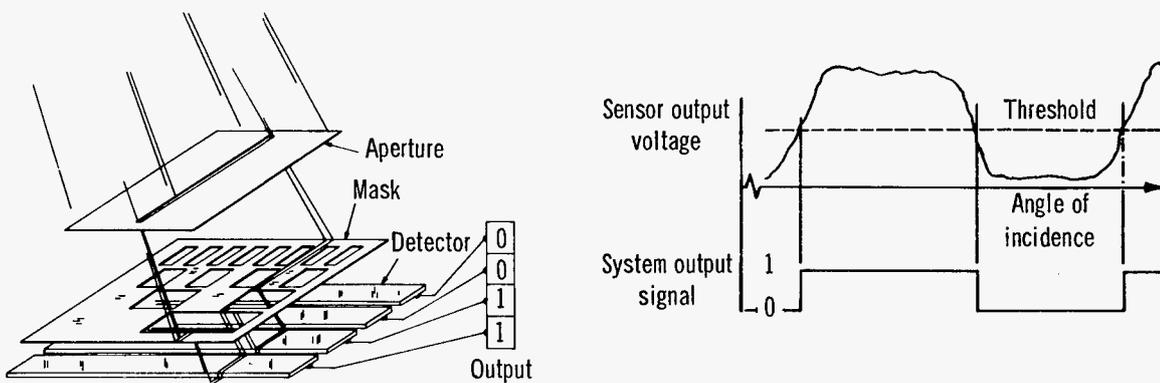


Figure 2.—Digital sun sensor.

In the simple one-axis analog sensor, the Sun's image illuminates two closely spaced photo-sensitive elements. The difference between the current outputs developed across the two elements becomes the sensor output. As the illumination of both elements becomes nearly equal, the sensor output approaches zero. This is the null point of the sensor.

In the digital sensor shown, the Sun is imaged as a line across an array of separate elements. Each element produces a 1 or 0 binary bit in the multichannel output, depending on whether light reaches the element through the mask and whether the sensor's output in each channel exceeds threshold values established in associated circuitry. The binary number assigned to the channel identifies the position of the light imaged on the array of elements.

Figure 3 presents a simple model of the functional elements of the sun sensor and facilitates consideration of the various hardware components that affect sensor operation. The functional elements of the sensor are the spectral filter, the spatial filter, and the radiation sensitive element.

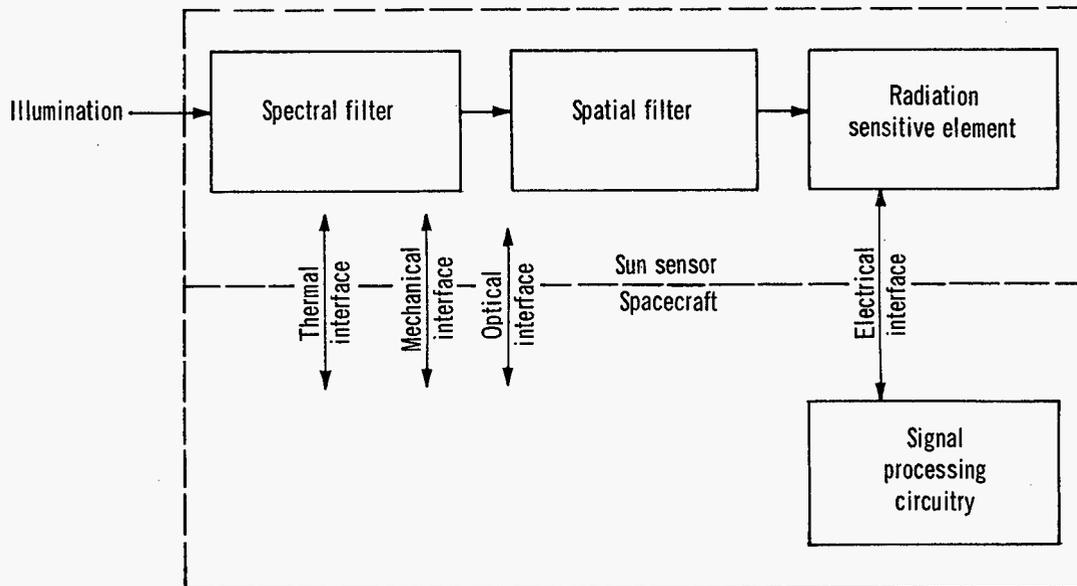


Figure 3.—Sun sensor functional elements and interfaces.

The spectral filters modify the incoming solar illumination, eliminating spectral bands where response from the radiation sensitive element is not desired. Ideally, the filters are selected to maximize useful information and minimize extraneous or misleading signals.

The spatial filter is the system of geometric relationships within, or external to, the sensor that enable interpretation of the direction vector between the system and the radiation source. This interpretation is accomplished in one of two ways. In one case, typical of digital sensors, the radiation arriving from a particular solid angle of space is compared to a

preset threshold value. The alternate method, typical of analog sensors, involves comparing the radiation arriving through a selected solid angle to that coming from adjacent regions within the sensor system's field of view. Individual sensors, distributed about the spacecraft structure, are often used to obtain spatial filtering by comparing radiation received through various solid angles with threshold values.

The radiation sensitive element converts the radiant energy passing through the spectral and spatial filters into electrical signals. The signals may be used to correlate experimental data for transmission to Earth, or used as the input information in the spacecraft's attitude control system. The processing or utilization of the sensor's output is beyond the scope of this monograph.

In some cases, the functional elements of the model can be identified with specific hardware components. In most cases, however, the characteristics of the hardware components are distributed among the several elements. For example, the intrinsic spectral response of the radiation sensitive element must be considered functionally as part of the spectral filter. Table I relates the model's functional blocks to the real components and their commonly specified performance parameters.

TABLE I.—Sun Sensor Functional Elements and Related Components

Functional block	Associated parameters	Hardware components
Spectral filter	Spectrum of source Element spectral response Optical transmission Angle of incidence	Radiation detector Filters Lens (optical elements)
Spatial filter	Proportional range Field of view Linear range Resolution Cross-coupling Data format	Shields Masks Blades Aperture Detector shape Lens Reticles
Radiation sensitive element	Sensitivity Noise current (or voltage) Bandwidth Load impedance Hysteresis Spectral response	Radiation detector Electronic interface

2.3 Flight and Test Experience

Although most sun sensors have performed satisfactorily in flight, some have displayed erratic behavior, and some have caused mission failures. Furthermore, few development or test programs have reached completion without encountering problems resulting in design modification, additional expense, or program delay. These problems will be categorized and illustrated.

2.3.1 Thermal Environment

The OAO sun sensors, mounted on thermally isolated appendages of the spacecraft, were required to survive several qualification cycles between -107° and $+72^{\circ}$ C. Some of the units subjected to these tests displayed radical changes in their operating characteristics. This indicated a noncatastrophic, but progressive failure mechanism. However, several sensors that failed this test developed cracks in the silicon detector: bimetallic action between the solder layer on the back of the detector and the silicon base material had generated sufficient stress at the low temperature extreme to fracture the cell. Corrective action included reduction of the thickness and area of the solder, and the substitution of a Kovar mounting substrate to achieve a better thermal expansion match between substrate and detector. Additionally, a program was initiated for the design and development of detectors capable of operation at extremely low temperatures. The merits of this program were demonstrated in the successful operation of sensors aboard OAO 2.

A similar problem was experienced during a program to develop a prototype sun sensor for an advanced Pioneer mission (ref. 8). In this case, however, the fracture of the silicon detector occurred at the high rather than the low extreme of a test cycle between room temperature and 150° C. The recommended solution was essentially the same as that described previously.

Relatively high sensitivity to temperature was exhibited by photo SCR detectors developed for the Pioneer 6 (ref. 9) with some units becoming totally inoperative at -45° C. Although the problem was reduced by modifications in the manufacturing procedure and bias circuitry (a thermistor was employed for bias stability), difficulties and high rejection rates continued throughout the program.

Mechanical parts tend to shift during thermal cycling because of stress relief and creep. The importance of proper mounting design was exemplified in a number of programs when sensor alinement suffered. In the OSO program, where alinement must be maintained through the thermal transient associated with solar occultation, the alinement procedure calls for alternate thermal cycling and adjustment until the measured alinement shift remains within tolerance.

Temperature extremes and gradients experienced during flights have occasionally exceeded initial expectations. On an early Aerobee rocket flight, an unanticipated angle of attack

resulted in overheating of a skin-mounted sensor and subsequent improper operation of the attitude control system. More recently, a digital sun sensor on an NRL satellite was subjected, during part of each orbit, to in-flight temperatures that exceeded design specifications. During these periods the sensor output was degraded and data bits were lost.

2.3.2 Radiation Environment

Radiation damage to sun sensors has occurred in several programs. A change in photosensitivity in units aboard OSO 1 resulted from the Starfish nuclear test; whether the damage was to the optics or the detector could not be determined from the telemetry data. A program was initiated to improve the radiation resistance of the sensors on later OSO missions, with particular attention to the selection of nonbrowning optical components and radiation hardened detectors. The sun sensors on recent OSO's employ cerium-doped glass, both in the lenses and the spectral filter, and n-on-p silicon detectors to improve radiation resistance (ref. 10).

Photo-SCR detectors aboard Pioneers 6 and 7 were incapacitated by radiation flux levels several orders of magnitude below the damage threshold level for silicon solar cells (ref. 9). The protective glass covers on the sun sensors aboard Pioneer 8 were increased in thickness from 0.02 to 0.10 in. to alleviate this problem.

Important data concerning space radiation and its effects on materials, particularly silicon diodes and glass, have been obtained from the flight experience of solar panels. The development of radiation resistant solar cells and cover glass has also benefited sun sensor development because comparable parts are used. Other radiation effects that may be significant in sun sensor design are the embrittlement of wire insulation, the discoloration of painted surfaces, and the bleaching and flaking of black anodized surfaces.

2.3.3 Handling and Contamination

The ability of dirt, dust, metallic particles, vapors, etc., to severely impair the performance of a sun sensor is generally well appreciated, and most sensors are assembled under clean-room conditions. However, contamination of the completed assembly may occur during spacecraft integration, test, and flight.

A sensor is not out of danger from rough handling and contamination after it is installed on the spacecraft. For example, the sun sensors of the Radio Astronomy Explorer were mounted on their respective solar panels before the start of the panel test program. Hence, the sensors were subjected to needless handling, and four out of the eight sensors were damaged because the test personnel were not made fully aware of the sensor's susceptibility to damage.

Nosecones of early Aerobee rockets were painted to aid in recovery operations. Later, when the rocket velocity was increased to achieve higher altitudes, the increased aerodynamic heating made the paint flow over the sensor ports, reducing the output by 75 percent and causing mission failure.

White paint covering the nonoptical surfaces of Lunar Orbiter's sun sensors outgassed during vacuum testing and clouded the quartz window of the test chamber. This problem was solved by vacuum cycling the coating until the outgassing was negligible. The paint was also found to be very sensitive to chipping, and procedures had to be developed to insure proper paint mixing, application, and part handling thereafter (ref. 11).

One of the sun sensors used on the Solar Pointing Aerobee Rocket Control System (SPARCS) was contaminated during a vibration test. When the contaminant was removed, some of the cement used to join the optical wedge to the aperture ring was also removed. This shifted the null point (by altering the spherical aberration of the optical system) and necessitated realignment (refs. 12 and 13).

These comprise only a small sample of the experiences that have caused most aerospace organizations that manufacture or use sun sensors to adopt special handling methods and procedures to reduce the risk of damage during spacecraft integration and test.

2.3.4 Vibration and Shock

Damage caused by vibration or shock has rarely been of a catastrophic nature. Most sun sensor components are small and rugged since the introduction of solid-state detectors. Generally, the principal effect of this environment is to alter sensor alignment.

2.3.5 Interference

Stray illumination (e.g., sunlight reflected from the Earth, Moon, or adjacent spacecraft surfaces) has caused problems on several missions. Lunar Orbiter's attitude control was confused by Earth and Moon albedo until control was transferred from the wide angle to the narrow angle (fine) sun sensors after initial Sun acquisition. The coarse sun sensors supplied yaw attitude information during off-Sun line operation. During certain portions of the orbit, the effect of moonlight was observable but did not degrade mission performance (ref. 11).

In the Mariner 5 mission, the effect of Earth albedo on the sun sensor was underestimated. This caused the lack of sun gate signal at Sun acquisition and delayed the next spacecraft event until the distance from the Earth was sufficient to diminish interference by reflected sunlight.

Similarly, the effect of Earth albedo on the Surveyor spacecraft's sun sensor was not properly assessed. The sensor had to be redesigned in this case to eliminate the problems.

Illumination interference on Nimbus 1 caused the spacecraft's solar panels to deviate by about 20° from their proper orientation with respect to the Sun. This problem was traced to a thermal shield that extended further into a sensor's field of view than had been anticipated because the planned circular orbit was not obtained. When the spacecraft attitude deviated by 1° or 2° from local vertical, the shield shadowed the sensor supplying orientation signals for the panels. The Sun shield was redesigned for later spacecraft so that attitude errors as great as 10° would not cause this type of interference.

2.3.6 Interface Problems

Mechanical and electrical difficulties have been experienced in the mating of sensors to the spacecraft.

A sun sensor was mounted in a rotated position on Pegasus 1 (ref. 14). This was classified as both a design failure and a human error because no keying of the sensor head was provided and because insufficient caution to assure proper orientation was exercised during installation and checkout.

On the first SPARCS rocket flight, the sun sensor pitch channel was cross-wired to the control system yaw channel, an error attributed to inconsistent nomenclature used by the sensor and control system designers for the pitch and yaw error signals (ref. 15). Because the prelaunch test procedures were not of the end-to-end type, the wiring fault was not discovered. On later flights, a special test was incorporated to insure that the proper attitude control jet operated when the corresponding sun sensor channel was stimulated.

During integrated systems testing of the Pioneer 6 spacecraft, it was found that a slow rate of illumination of the sun sensors resulted in excessive leakage current through the photo-SCR detectors. This reduced the supply voltage to so low a value that when the SCR switched to the conductive state, the generated pulse was inadequate to trigger the logic circuits in the orientation control electronics. This problem was solved by the addition of a resistor-capacitor network (ref. 9).

2.4 Sun Models

Available solar radiation models covering the spectral region from the ultraviolet to the infrared are sufficiently detailed for all but the most exacting sensor design requirements. For design problems where sensor accuracy does not exceed approximately 1 arcmin, an appropriate model is described in the NASA design criteria monograph on solar radiation (ref. 16). Additional material may be found in the survey of literature on the spectral

distribution of solar radiation (ref. 17), but this should be updated by the recent corrections to the solar constant given in reference 18. When better than 1 arcmin accuracy is required, the material contained in references 23 and 24 will be helpful.

2.5 Albedo Models

A method for estimating the effects of planetary albedo on sun sensor performance and typical values for the albedos of the planets are given in reference 17. More detailed information on the albedos of Mercury, Venus, and Mars is contained in reference 18. Together with references 19 and 20, they provide a basis for assessing the effects on sensor performance of the sunlight reflected by Earth, Mercury, Venus, Mars, and the Moon.

2.6 Sun Sensor Testing

Test methods are highly dependent on the sensor's physical characteristics, its performance specifications, and its function in the spacecraft system. Testing is not presently as advanced as the state of the art in design. The major difficulty lies in obtaining adequate solar simulation, particularly in simulated flight environments. Definitive sun sensor tests have rarely been possible after the sensor has been mated to the spacecraft. This is especially true for multiple sensors requiring a large, uniform beam for simultaneous activation.

Within present technology it is impossible to simultaneously simulate the spectral radiance and angular subtense of the Sun as viewed from above the Earth's atmosphere. Thus, data from sensor testing always require careful extrapolations to the operating conditions of space flight. Sensor performance testing has been accomplished using the following approaches:

- (1) Outdoor testing using the Sun viewed through the Earth's atmosphere
- (2) Indoor testing using mirror systems to relay sunlight to a test area
- (3) Indoor testing using artificial light sources

The following disadvantages exist in the use of the Sun as the test source:

- (1) Effect of the atmosphere on the intensity and spectral content of the sunlight
- (2) Effect of atmospheric refraction
- (3) Bright sky background
- (4) Reflection from surrounding structures
- (5) Dependence of test schedules on weather conditions

Where accuracy of 30 arcsec or better is not required, the first two effects may not be important. When assiduous effort is applied, an alignment accuracy of 10 to 30 arcsec and a calibration accuracy of 2 to 5 percent can presently be achieved using natural sunlight.

An artificial illumination source closely approximating outer space sunlight is a blackbody source at approximately 6000° K. Portions of the illumination produced by mercury-xenon and carbon arc sources are close to this temperature; however, they are of very limited extent and tend to vary in position. Other undesirable features of these sources are the large excursions in the radiometric properties of the arc and the spectral spikes; i.e., narrow spectral regions of abnormally high or low radiance that deviate from the solar spectrum. Finally, the carbon arc is a dirty source that presents a serious risk of contamination to both the sensor and the test.

Other solar simulators that have been used include tungsten lamps, tungsten blackbody cavities, and hypereutectic graphite sources. These sources have good stability and uniformity but fall far short of the radiance level of sunlight (about 1/60 to 1/100 in the 0.6 μ to 0.9 μ spectral region). The ribbon filament is considered the most acceptable configuration of the tungsten sources. Still another source used frequently to test the assembled spacecraft is a system of small lamps located near the sensors. The lamps are in some cases automatically sequenced to stimulate vehicular attitude change; e.g., roll. In these "go/no go" type tests, the duplication of solar characteristics is not as important as assuring that the interdependent systems are functioning properly.

Qualification tests of sensors having accuracy requirements of 1 arsec or better demand highly sophisticated test facilities. Special air conditioning and isolation from vibration must normally be employed, in addition to the most stable illumination sources. Clean-room techniques are frequently used.

2.7 Summary

Sun sensor flight experience has been favorable, and very few in-flight failures have occurred. In those cases where anomalous behavior was observed, the principal causes were inadequate consideration of the environmental ranges, unwanted illumination from sources other than the Sun, and inadequate interface design.

In contrast to flight experience, development programs have been plagued with problems that resulted in increased costs and delays. Difficulties encountered have included handling and contamination damage, inability of available components to meet previously established specifications, unanticipated behavior of components subjected to environmental tests, fabrication problems, inadequate consideration of testing and calibration methods at the design level, and limitations of test equipment and facilities.

3. CRITERIA

Spacecraft sun sensors shall be designed to provide signals indicating the angular orientation of the Sun relative to a specified reference axis under all specified mission conditions. The

design of the sensor and the associated systems of the spacecraft must take into account all factors that can degrade reliable performance. These factors include the external environment, the various interfaces between the sensor and the spacecraft, possible interference sources, the solar characteristics, and the sensor's functional characteristics. Demonstration shall be made, by a suitable combination of analytical and experimental studies, that the sensor design is entirely suitable for its intended mission.

3.1 Evaluation and Control of Sensor Inputs

3.1.1 Selection of a Solar Model

The characteristics of the Sun that affect the operation and performance of the sensor shall be identified, and a solar model suitable for design, analysis, and testing shall be established. The model shall treat the following:

- (1) Angular subtense and variations over the spectral range of interest
- (2) Spectral irradiance and variations over the spectral range of interest
- (3) Spatial characteristics such as limb darkening, sunspots, and prominences
- (4) Spacecraft trajectory parameters and their effects upon solar parameters

3.1.2 Design of Internal Sources

Some sensors incorporate an internal source of light for calibration purposes or for integral autocollimators. In such cases, these sources shall be examined to assure that sensor performance will not be degraded by faulty design. Consideration should be given to certain characteristics:

- (1) Reflection
- (2) Dispersion
- (3) Light leakage
- (4) Anomalous behavior of the source, including filament failure or degradation

3.1.3 Control of Stray Illumination and Shadowing

Test and analytical procedures shall demonstrate that neither unwanted illumination nor shadowing compromise sensor performance for any anticipated spacecraft orientation or for any possible location of deployable or movable appendages or other interfering bodies. Mission planning of trajectory and schedule shall be reviewed to assess risks of shadowing caused by natural phenomena or mission anomalies.

The effect upon sensor operation of illumination by other than direct sunlight shall be evaluated. Such illumination sources include:

- (1) Sunlight reflected from the Earth, Moon, or planets (albedo)
- (2) Sunlight reflected from spacecraft surfaces or appendages, such as booms, antennas, and solar panels
- (3) Sunlight reflected from sensor parts, such as shields, baffles, lenses, internal surfaces, and mounting hardware

The effects of partial or total solar occultation by spacecraft parts should be examined. Shadowing may occur from –

- (1) Spacecraft parts, including boom, antennas, solar panels, and support structures
- (2) Planets, natural satellites, and asteroids
- (3) Off-nominal trajectories

3.2 Sun Sensor Functional Characteristics

The sun sensor design shall be consistent with the system performance specifications, the mission objectives, and the selected test program.

The characteristics of individual sensor components shall be examined for compatibility and for assurance that the properties of one component do not unnecessarily constrain the selection of other components.

The design must take into account the susceptibility of each component to degradation in operational and test environments to insure reliable performance throughout the mission.

3.2.1 Spatial Filter

A reference coordinate system shall be established and used as a basis for specifying the characteristics of the spatial filter. It shall be demonstrated that the spatial filter provides angular subtense and resolution required for the mission and that overall sensor reliability will not be degraded by changes in the characteristics of the spatial filter components, such as lenses, mirrors, masks, shields, and associated mounting structures.

3.2.2 Spectral Filter

It shall be demonstrated that the overall spectral filter characteristics are compatible with sensor performance objectives and specifications and will not be degraded by changes in the characteristics of components, such as lenses, filters, and photosensitive semiconductors.

3.2.3 Radiation Sensitive Element

It shall be demonstrated that overall sensor effectiveness will not be degraded by changes in the characteristics of the photosensitive detector. The characteristics of the radiation sensitive element shall be compatible with the following constraints:

- (1) Magnitude of the incident radiant energy
- (2) Required sensor bandwidth
- (3) Allowable noise-equivalent-angle (see appendix)
- (4) Linearity requirements

3.3 Interfaces With Spacecraft

3.3.1 Spacecraft System Interferences

Interface conditions should be specified, if possible, by interface definition documentation. It shall be demonstrated that both the sensor, and the system to which the sensor interfaces, will perform satisfactorily when the specified conditions are met.

It shall be demonstrated that the sun sensor system will not be adversely affected by the presence or the operation of other spacecraft systems. Consideration must be given to several factors:

- (1) Electrical interfaces with the spacecraft electronics and power system by means of ground currents, electrostatic discharges, magnetic fields, and electric fields (radiofrequency, audiofrequency, impulse, etc.)
- (2) Mechanical interfaces—including stability, rigidity, and effects of mounting and demounting, alinement and access, and thermal expansion
- (3) Thermal interfaces between sensor and spacecraft (heat transfer by conduction or radiation)
- (4) Radiation interface with radioactive devices (nuclear power or calibration sources)
- (5) Interfaces with the sensor test equipment

3.3.2 Alinement to the Reference Axes

Provisions shall be made for the adjustment of sensor output relative to an established coordinate frame on the spacecraft. Adjustment techniques, whether electronic, optical, or mechanical, or combinations of these methods, must provide stability of alinement commensurate with mission requirements.

3.4 Environmental Effects

Environmental specifications should include conditions to be experienced during handling, storage, shipment, assembly, test, prelaunch, launch, and flight. The capability of the equipment to operate or survive during or after exposure to appropriate combinations of these environments shall be demonstrated.

A model or specification, sufficiently accurate for evaluation purposes, shall be established for each of the following environmental factors:

- (1) Radiation (including thermal, ultraviolet, X-ray, and particulate)
- (2) Vibration, shock, and acceleration
- (3) Micrometeoroid flux
- (4) Contamination (including condensation, debris, fingerprints, etc.)
- (5) Temperature
- (6) Vacuum
- (7) Salt spray
- (8) Water vapor
- (9) Electromagnetic field

3.5 Evaluation of Performance

A set of measurable characteristics shall be established to define the test data by which the evaluation of the sensor design and performance in the operational ranges of environment can be made. Depending on the sensor design, this set will include:

- (1) Physical factors; e.g., weight, size, configuration, power requirements, and reference coordinate system(s)
- (2) Lifetime and reliability
- (3) Performance factors (see appendix), such as angular resolution, absolute accuracy, stability, field of view, transfer function, and output format

3.5.1 Illumination Source for Testing

An assessment shall be made of the characteristics of the illumination source(s) required to verify proper sensor performance. The characteristics of the established solar model should be compared to those of the source(s) used for test illumination. The effect of differences between the solar model and the test source should be evaluated for each of the following:

- (1) Angular subtense and collimation
- (2) Spatial factors, such as geometric accuracy and stability, edge characteristics, and beam uniformity and stability

- (3) Spectral characteristics over the range of interest
- (4) Variations in background illumination
- (5) Noise generated by the test source
- (6) Mechanical and optical effects of atmospheric pressure

Additionally, it should be assured that the test source will not present an environmental hazard, such as excessive heat or contamination.

3.5.2 Testing

Tests shall be performed that sufficiently establish that the sensor performance parameters are within specifications under all expected operating conditions. The test procedures shall not introduce extraneous operating conditions or environments. The test plan should provide, where applicable, for the following:

- (1) Component and subassembly tests
- (2) Calibration test
- (3) Alinement test
- (4) Testing on the spacecraft, or in a fixture that duplicates that portion of the spacecraft within the sensor's field of view
- (5) Sensor operation in simulated flight environment
- (6) End-to-end field tests
- (7) Prelaunch "go/no go" test

3.5.3 Calibration

The sensor output calibration under flight conditions shall be evaluated by extrapolation from test results. The extrapolation procedure used should be compatible with the accuracy requirements established for the mission.

4. RECOMMENDED PRACTICES

The mission requirements, interfaces between the sensor and other equipment, characteristics of possible interferences, and the range of environmental stresses that may be imposed on the sensor should be evaluated prior to the design of sun sensors for use in spacecraft. Early development and consideration of alternative designs is recommended. The configuration that meets budget and schedule constraints and does not require unrealistic capabilities of its components should be selected. The selected design should have inherent testability to obtain the necessary performance data at all levels: component, subassembly, assembly, and when installed in the spacecraft. Testing should be explicitly and carefully considered in program planning and scheduling.

4.1 Evaluation and Control of Sensor Inputs

4.1.1 Solar Characteristics Model

A principal constraint on the specification of a solar model is that it be selected to insure compatibility with the design concepts chosen for the sun sensor. This model will provide the radiation definition for the analysis of sensor performance and a basis for the selection of a test source. The following guidelines should be considered in selecting the type and detail of the model.

- (1) A model comprised of a 6000° K blackbody subtending a solid angle of 32 arcmin is generally adequate for sensors having spatial filters with a preset threshold value and that will be used in missions near 1 A.U.
- (2) For analog sensors used in pointing applications, it may be necessary to include limb darkening, spatial detail of broad (e.g., sunspots) or narrow (e.g., solar flare) spectral features, and fine spectral detail. Limb darkening affects the linearity of the sensor transfer function, whereas spatial detail of spectral phenomena can affect both linearity and accuracy. Spatial detail of the spectral features is important, and may be critical, for sensor accuracy of the order of 1 arcsec or spectral response that is very narrow; e.g., less than 1000 Å

Because the Earth's orbit is eccentric, the mean solar intensity and apparent diameter vary over the course of the year by approximately 6 and 3 percent, respectively. This factor can be an important consideration in certain designs.

4.1.2 Design of Internal Illumination Sources

Sun sensor designs that have internal light sources for calibration, "go/no go" tests, or other operational purposes, should be carefully examined for possible leakage paths by which light may inadvertently reach the detector. Such paths include reflections, scattering, edge illumination of optical parts, and flaws in opaque elements. Special attention in design and in quality assurance provisions is needed to guard against the introduction of this type of defect.

4.1.3 Control of Stray Illumination

The planned spacecraft trajectory and attitude and the sensor's field of view should be evaluated for the possibility of unwanted illumination from the Earth, Moon, and other planets. A narrow field of view substantially reduces the likelihood of albedo interference; however, performance limitations associated with this measure should be examined. Where wide-angle sensors are employed for initial Sun acquisition on Earth-orbiting spacecraft,

Earth albedo problems can be minimized if acquisition is accomplished just after the spacecraft leaves the Earth's umbra; control is then transferred to narrow-angle sensors. An alternative approach using the combination of a wide-angle coarse sensor and a narrow-angle fine sensor in a two-axis null-pointing application is presented in reference 25.

Reflections from within the sensor and from external spacecraft parts can become a more serious problem than planetary albedo because the surfaces subtend large angles and may exhibit brightness comparable to that of the Sun. Throughout the design, fabrication, and assembly of sensor components, careful attention should be given to the potential problem of internal sensor reflections. The sensor test program should detect any unexpected internal reflections so that corrective measures may be applied.

The location of the sensor on the spacecraft should minimize reflected light. The specific solutions will usually require some compromises or tradeoffs: reflection may be completely eliminated by mounting the sensor on the outer edge of a deployed solar panel, but stability and reliability may be degraded.

Where possible, use external shields to intercept light from sources whose position relative to the sensor is known. The most effective shields employ highly absorbing, specularly, rather than diffusely, reflecting materials. It is preferable to reflect the light to another part of the shield or back into space rather than to permit diffuse reflection that would almost certainly cause some of the light to arrive at the sensor (ref. 26).

The design efforts for spacecraft configuration, surface properties, and sun sensors should be formally coordinated to insure safeguarding of the sensors from reflections from external surfaces. Discovering such reflections in tests is not always possible; the small diameter of the test light beam and the impracticality of testing with the spacecraft in the deployed configuration may preclude such detection.

4.1.4 Control of Shadowing

The possibility that some appendage of the spacecraft will cast a shadow on the sensor increases with the field of view of the sensor. Sensors should be located so that they will not be shadowed. When this is not possible, the consequences to the mission should be evaluated. For some missions it may be necessary to employ an additional sensor, similarly oriented and sufficiently displaced from the primary sensor to be in the clear. Auxiliary switching sensors can be used to detect an impending shadow and transfer the input channels of the receiving system to an unshadowed sensor (ref. 27).

If a sensor will be occulted during the deployment of an appendage, the effect of failure to achieve complete deployment should be investigated. It is good practice to clearly indicate in drawings the cone angles that must be kept free of obstructions. There is no substitute for clear communications between spacecraft configuration and sensor design groups for dealing with problems of sun sensor shadowing.

4.2 Sensor Characteristics

Sensor designs have utilized a wide variety of materials, physical forms, and principles of operation; however, recommended practices can be appropriately organized according to the functional blocks of the sun sensor model described in section 2.2.

4.2.1 Spatial Filter

The primary factors that must be considered in the design of the spatial filter are the desired field of view and transfer function of the sensor; performance factors, such as absolute accuracy and short-term drift; minimization of the sensor's susceptibility to spurious illumination; and requirements for the coordinate reference system. Secondary factors for consideration are hardware availability, physical size, and susceptibility to contamination.

The following are general recommended practices:

- (1) The structure of the sensor should have the same coefficient of thermal expansion as the lens material. The lens should be mounted so that thermal expansion and contraction does not change angular alignment. One useful technique is the use of flat, hardened, polished pads on the sensor bearing against an optically flat mounting surface (ref. 13). Special techniques of this type are needed to achieve absolute accuracies better than about 1 arcmin.
- (2) For accuracy requirements less stringent than 1 arcmin, more conventional mounting practices may be used. However, mounting techniques that apply large mechanical stresses to the critical components (e.g., staking or spinning optical components into housings) should be avoided.
- (3) Deposition of aluminum or chromium reticles on optical substrates should be considered to provide straightness and edge deviation tolerances of 50μ inches. This measure is imperative if the dimensional accuracy and stability of a blade, reticle, or knife edge will be critical to system performance. Even when extreme accuracy of an edge is not necessary, the deposition technique has the advantage of being less susceptible to damage, although pinhole defects must be guarded against.
- (4) Coatings applied to a transparent substrate to produce the slit aperture and code mask for digital sensors should have an optical density of 3 or more, be stable in the sensor's test and operating environments, and be free from pinhole defects. If glass is the substrate, it should be homogeneous; free from bubbles, striations, and stresses; and have a known and controlled index of refraction. These characteristics are critical to digital sensors requiring 1 arcmin or better accuracy; the best available quality optical grade of fused silica is recommended.

An approximate transfer function should be derived using a geometrical analysis supplemented with test results from breadboard systems. The spatial filter transfer function for sensors with a proportional range of 5° or larger is conveniently determined by analytical methods. This is not so for sensors with a smaller proportional range because diffraction,

source nonuniformity, and spherical and chromatic aberration can no longer be considered as second-order effects. Extension of the analysis to include such effects is possible but time consuming. Analysis is further complicated by the common practice of defocusing imaging-type sensors to improve their linearity. Final verification of the actual transfer function should be obtained from tests made on a prototype sensor.

4.2.2 Spectral Filter

For most sensors, the required spectral response characteristics can be obtained by incorporating into them either thin film dielectric, or absorption-type filters. Spectral transmission properties of absorption-type filters are slightly temperature dependent, vary from batch to batch, and may vary from point to point over a typical sample by as much as 10 percent; generally, thin film dielectric filters are more stable and uniform, allow more freedom in bandpass specification than is possible with absorption-type filters, and have spectral characteristics that vary with the angle of incidence.

The spectral filter should be designed to shape the spectral response of the sun sensor to enhance test ability. The sensor's primary spectral response should cover the region where the test source spectrum can be made to correspond closely to the solar spectrum. When this is the case, testing will be easier and the probability of introducing large calibration errors will be reduced. If the sensor is to be tested in sunlight at the Earth's surface, the spectral filter characteristic should correspond to a window region of the atmosphere (ref. 28). Spectral filtering should be applied to tungsten test sources to produce a close spectral match to the Sun over wavelengths from 0.6μ to 0.9μ (ref. 12). A bandpass filter of this range in the sensor can minimize the temperature sensitivity of the silicon detector spectral response.

The disturbing effects of the sky background should be considered in outdoor testing, particularly if the sensor has a wide field of view. A short wavelength cutoff filter can minimize these effects. When the test illumination is either carbon arc or mercury xenon arc, particular care should be taken so that spectral spikes in these sources do not cause errors in calibration and alignment.

4.2.3 Radiation Sensitive Element

Important characteristics of photosensitive devices used in sun sensors are compared in table 2. The most commonly used and recommended detector for spaceborne sun sensors is the silicon solar cell. It has the comparative advantages of good linearity over a wide range of illumination levels, a short time constant that is independent of light level, excellent temperature stability in the short-circuit current operating mode, a good spectral match to the Sun and typical artificial sources used in testing, excellent reliability, a wide operating temperature range, low noise, ruggedness, and good radiation damage resistance, if n-on-p-type high resistivity material (e.g., 30 ohm-cm) is used.

TABLE 2.—Comparison of Detector Characteristics

Detector	Spectral range, μ	Internal gain	Linearity (angular)	Temperature stability	Reliability	Operating temperature range, $^{\circ}\text{C}$	External power required	Radiation resistance	Time constant
Silicon solar cell	0.3-1.0	1	Good	Very good	Very good	-80° $+100^{\circ}$	None	n-p: good p-n: fair	$\sim 30 \mu\text{sec}$
Silicon photo-transistor	0.3-1.0	~ 30	Fair	Fair	Good	-20° $+85^{\circ}$	Low voltage	Fair	$\sim 30 \mu\text{sec}$
Photo-SCR	0.3-1.0	NA	NA	Poor	Fair	-20° $+85^{\circ}$	Low voltage	Fair	
Cadmium sulfide	0.4-6	$\sim 10^3$	Fair	Poor	Good	-20° $+85^{\circ}$	Low voltage bias	Good	$\sim 100 \text{ msec}^{\text{a}}$
Cadmium selenide	0.4-7	$\sim 10^4$	Fair	Poor	Good	-20° $+85^{\circ}$	Low voltage bias	Good	$\sim 50 \text{ msec}^{\text{a}}$
Photomultipliers (S11, S20, S1)		$\sim 10^6$	Fair	Poor	Fair	-200° $+85^{\circ}$	High voltage ^b	Good	$\sim 20 \text{ nsec}$

^aTime constant varies with light level.

^bCorona problem.

The following recommendations apply to silicon solar cell detectors:

- (1) The detectors should operate into an impedance sufficiently small that the diode current will be low compared to the output current. Operation of the cell in the short-circuit current mode affords much better temperature stability and linearity.
- (2) Where thermal extremes fall outside the range of -20° to $+85^{\circ}$ C, the cells should be mounted on a substrate with matching thermal coefficient (Kovar, glass, etc.) using a minimum solder thickness.
- (3) The cell impedance should be increased by using a mesa etched unit. The higher impedance minimizes current offsets caused by offset voltage at the amplifier input.
- (4) Where high radiation levels are expected in the mission, n-on-p cells should be preirradiated to reduce degradation. Reference 25 provides an example of this treatment.

The radiation sensitive elements of sun sensors have a varying response depending on where the light falls on the detector surface. Where such behavior may be detrimental to sensor performance, the detector should be located so that a large portion of its surface is illuminated. Motion of the illuminated region across the detector surface should be minimized.

The temperature sensitivity of phototransistors and photo-SCR's has caused major problems in the past and their use is not recommended. Photomultipliers present corona problems and their use should be limited to those applications where other detectors are unsuitable. Cadmium sulfide and cadmium selenide detectors have high-temperature sensitivity and "memory" (i.e., the responsivity of the detector is a function of past exposure to illumination). These detectors should only be used in those applications where temperature stability and accurately predictable response are not of paramount importance.

4.3 Interfaces With the Space Vehicle

The interfaces between the sun sensor and the other spacecraft equipment will frequently introduce mutual design constraints. For example, magnetic cleanliness requirements may preclude the use of certain commonly used materials such as Kovar and certain types of stainless steel.

4.3.1 Electrical Interface

The electrical interfaces between the sensor and the spacecraft electrical system (power supplies, control system input, telemetry, etc.) should be given thorough consideration

during the design. Commonly occurring difficulties involving the electrical interface between the output of the electronics and the receiving system can be avoided by –

- (1) Providing a positive means for identifying the proper signal channel or signal polarity
- (2) Carefully matching impedances
- (3) Assuring correct signal levels
- (4) Minimizing noise on the power bus
- (5) Providing protection against transients, overloads, and shorts

Transients, overloads, and shorts can pose especially serious problems. Whenever monitoring signals are taken from a sensor output, it should be assured that a failure or short circuit in the monitoring equipment will not render the sensor output ineffective for critical operations.

4.3.2 Electromagnetic Interference

Major sources of electromagnetic interference (EMI) on the spacecraft are antennas, antenna feed lines, solenoids, relays and their power supply lines, transmitters, electric motors, and gyros. Sun sensors are particularly sensitive to such interference in the radiofrequency range because the need for an optical window makes complete shielding difficult; silicon solar cell detectors act as a demodulator for radiofrequency interference (RFI), resulting in bias or a spurious signal; and long connecting leads are required when the sensor and its associated electronics cannot be placed in adjacent positions.

The following practices are useful in reducing RFI problems:

- (1) Locate at least one stage of amplification and low impedance line drivers in or near the sensor assembly
- (2) Use shielded leads of minimum length
- (3) Use good grounding techniques
- (4) Locate the sensor and its leads in areas that are relatively free of high-intensity radiofrequency fields. Locations in the side lobes of antennas should be avoided.

4.3.3 Thermal Interface

The thermal interface between the sensor and the surrounding structure is a major factor contributing to the operating temperature of the sensor. When the sensor must be thermally isolated from the structure (i.e., highly conductive paths are precluded), an analysis will be necessary to determine the temperature variations of the sensor. If the analysis indicates that the expected temperature extremes will compromise reliability, assessments should be made of the possible tradeoffs involving relaxation of thermal isolation requirements and sensor design changes. (See sec. 4.4.3.)

4.3.4 Mechanical Interface

It is vital that the designer ascertain the compatibility of the mounting surface with the specifications of the sensor. The following surface characteristics become increasingly important as the absolute accuracy or stability requirements become more stringent:

- (1) Flatness
- (2) Surface finish
- (3) Rigidity
- (4) Thermal expansion
- (5) Thermal conductivity
- (6) Hardness
- (7) Creep characteristics
- (8) Corrosion resistance

Practices that have proved useful in mechanical interface problems include hard anodizing of the sensor mounting pads (this technique is sometimes incompatible with thermal and EMI interface requirements), counterboring the mounting holes, use of three point mounts, and avoidance of mounting techniques that stress the sensor at points other than the mounting pads. Adequate clearance for removing the sensor from the mounting should be provided.

4.3.5 Alinement of the Sensor

The required alinement accuracy and the procedure for its accomplishment must be established early in the design of the sensor system. The degree of correspondence required between the sensor reference coordinate system and a coordinate system associated with the spacecraft will determine the magnitude of the alinement problem.

The objective of the alinement procedure is to orient the null axis of the sensor in the desired position with respect to the reference coordinate system of the spacecraft. Generally, this should be done indirectly because of the inconveniences in providing necessary tests and adjustments after the sensor is mounted on the spacecraft. Essentially, the null axis of the sensor is first determined relative to mounting determinants of the sensor housing. Then, the mounting position on the spacecraft is established or modified so that the alinement objective is achieved when the mounting takes place.

The optical axis of an analog sensor ideally would be determined by spinning the sensor in the light beam and adjusting the spin axis until zero modulation in the output is noted. In practice, however, the "plunging" technique is generally adequate. Here, the sensor is mounted on a test platform so that it can be positioned alternately in two orientations rotated 180° from each other. The apparent null axis for each orientation is noted and the bisector of the included angle is taken as the true null axis.

In many cases the absolute accuracy requirements are within the range attainable by mechanical mounting techniques without adjustment (1° to 5°), and good mechanical design practices are sufficient. Care should be exercised when sensors or their subassemblies are sensitive to rotation; it is good practice to provide keyed mountings in cases where alinement errors or operational difficulties could result from mounting the sensor in a rotated position.

When simple mechanical mounting procedures are not sufficient, the following techniques should be considered for achieving alinement:

- (1) Mounting surface lapping techniques
- (2) Structural deformation
- (3) Adjustable mechanical wedges
- (4) Adjustable optical wedges
- (5) Lockable ball and socket
- (6) Jack screw adjustment
- (7) Shimming
- (8) Adjustment (by bending or cutting) of external shields or blades
- (9) Electronic bias

Where absolute accuracies of the order of 1 arcmin or better are required, the use of surface lapping, optical wedges, and electronic bias yield more stable and accurate results than the other techniques.

4.4 Environmental Design Considerations

The flight, launch, prelaunch, and test environments to which the sensor will be exposed should be accounted for in the design to minimize the risk of performance degradation. Table 3 presents a summary of the environmental hazards to sun sensors.

Sun sensor specifications generally contain definitive information on specific environmental factors for launch and flight conditions including shock and vibration, radiation, contamination, micrometeoroid flux, temperature range, etc. However, environmental considerations should be extended to cover conditions of manufacturing and assembly, shipping and handling, and inspection and testing.

The prelaunch environment often represents the limiting case for environmental hazards. For example, it is not unusual for a sensor to experience higher shock or vibration levels during manufacturing or transit than during qualification testing or launch. The contamination environment experienced during handling, inspection, and manufacturing is almost always more severe than after launch. Constraints on the allowable exposures in prelaunch environments should be established and enforced to insure against accidental damage to the sensors.

TABLE 3.—Environmental Hazards to Sun Sensors

Sensor function			Hardware component	Environmental factor							
Spatial filter	Spectral filter	Radiation sensitive element		E=near Earth P=planetary	UV radiation	X-rays	Contamination	Particulate radiation	Vibration and shock	Thermal extremes	Micrometeoroids
X	X		Optical elements: lenses, mirrors, prisms	E	1	2	1	3	1	1	3
				P	1	2	1	1	1	1	1
X			Masks, blades, knife edges	E	4	4	1	4	1	1	4
				P	4	4	1	4	1	1	3
X			Shields	E	4	4	3	4	1	1	4
				P	4	4	3	4	1	1	4
X			Mounting structures and surfaces.	E	4	4	1	4	1	1	4
				P	4	4	1	4	1	1	4
	X		Spectral filters	E	2	3	1	3	1	1	3
				P	2	3	1	3	1	1	3
	X	X	Solid-state detectors: silicon, CdS, etc.	E	3	2	1	2	1	1	3
				P	3	2	1	1	1	1	3
			Temperature control surfaces	E	1	3	1	3	4	3	2
				P	1	3	1	3	4	3	2

1: severe; 2: mild; 3: second order; 4: no effect.

4.4.1 Radiation Environment

The severity of the radiation environment depends upon the trajectory and orbit parameters and, occasionally, upon other onboard equipment such as radioisotope thermoelectric generator (RTG) power supplies. Sensors flown in sounding rocket flights and low-altitude Earth orbits are not strongly affected by solar and cosmic radiation. However, when the mission plans call for considerable time within the natural or artificial radiation belts or near the Sun (closer than about 0.4 A.U.), resistance to radiation damage must be included in the sensor's design.

The parts of a sensor most susceptible to radiation damage are the detectors; optical elements; and, possibly, the surface finishes. The n-on-p type of silicon detector is more resistant to such damage than the p-on-n type (ref. 29). Cerium doping of optical elements should be considered; this technique has been used successfully to reduce radiation damage to optical elements (ref. 30).

Preirradiation of sensitive components is a beneficial technique that makes use of the fact that effects of the damage mechanisms tend to be exponential. The preirradiation dosage should be equivalent to that expected during the mission and the resulting degradation accounted for in the design. Subsequent degradation during flight will then be small (ref. 25).

4.4.2 Vibration and Shock Environment

The vibration and shock environment for sensors and related components can be significantly reduced by --

- (1) Using special handling packaging and shipping techniques
- (2) Providing special storage, assembly, and test areas and employing padded surfaces where necessary
- (3) Developing manufacturing processes that minimize shock or vibration levels
- (4) Educating involved personnel

It is recommended that the assembly program be arranged so the sensor is not subjected to unnecessary handling or exposures. For example, if a subassembly test program requires a mounted sun sensor, a temporary substitute unit should be used, if possible.

4.4.3 Thermal Environment

Extremely low temperature (-40° C and below) can cause sufficient mechanical stresses to damage or destroy sensor components. When such temperatures are anticipated, it is often practical to incorporate design provisions to prevent the exposure. Heaters have been used

for this purpose at the expense of battery power during the satellite night. By coupling the sensor to a spacecraft structure having a large thermal inertia, extremely low temperatures can often be avoided.

High temperatures usually reduce the sensitivity of the detector and increase the noise level. If the sensor is exposed to temperatures higher than approximately 85° C, mechanical problems similar to those encountered at low temperatures should be anticipated. Additionally, the high-temperature behavior of potting compounds, optical cements, coatings, etc., should be examined for outgassing, creep, loss of adherence and discoloration.

4.4.4 Contamination Environment

In-flight contamination can be reduced by special covers over the sensor apertures, but caution must be exercised to avoid stray reflection, interference with other equipment, or excessive reduction in overall reliability. Manned spacecraft have special contamination problems caused by waste disposal and outgassing of the heat shield.

Much of the contamination problem prior to launch can be effectively eliminated through special handling or cleaning procedures. Improper cleaning, however, can aggravate the problem, either because of residue or because of damage to delicate components.

4.4.5 Meteoroid Environment

For mission durations up to 1 yr in a near-Earth orbit, and for sounding rocket applications, meteoroid damage is not a serious problem. For much longer missions or other trajectories, meteoroid environment models (refs. 31 and 32) should be used to develop an estimate of the risk of such damage to sensors.

The sensor parts most likely to be damaged in such a way as to result in degraded sensor performance are exposed optical components and delicate mechanical components (e.g., masks and apertures). Removable covers can be employed in some cases to minimize meteoroid exposure; however, their use may compromise overall reliability.

Where damaging exposure is anticipated, the use of unprotected or bare silicon cells in sun sensors should be avoided. The effects of meteoroid damage on system performance can be diminished to a large extent by incorporating damage tolerance in the design. For example, an imaging type spatial discriminator employing a single lens for both control axes is more damage resistant than a similar nonimaging type. This is because spalling, which changes the scattering and transmission of the optical element, tends to affect both detectors of the imaging sensor equally. The damage in the nonimaging sensor, on the other hand, usually affects the energy reaching only one of the detectors, and a null shift results.

4.5 Testing and Calibrating Sun Sensors

A testing plan should be developed concurrently with sun sensor design to insure testability of components and the assembly. The plan should specify test procedures, equipment requirements, and any environmental controls necessary to obtain repeatable test results. The following paragraphs give details of planning considerations and practices related to illumination for test and calibration.

4.5.1 Test Illumination Source

The selection of an illumination source for sensor testing can have a significant effect on both the design of the sensor and the subsequent costs and schedules of the development program. It is imperative that the limitations and potential difficulties peculiar to the various illumination source options be recognized early in the program and properly assessed. Within the present technology it is impossible to simultaneously simulate the spectral radiance and angular subtense of the Sun as viewed from above the Earth's atmosphere. Thus, sensor testing always involves compromises and extrapolations of data to the operating conditions of the planned mission.

4.5.1.1 Natural Sunlight

Sunlight can be used to test sun sensors at the Earth's surface although the following factors impose serious constraints that should be evaluated for proposed applications.

- (1) Meteorological conditions may adversely affect testing schedules
- (2) Atmospheric effects, such as differential refraction and image jitter and flicker, limit the usefulness of the technique under the best atmospheric conditions to applications where alignment accuracies need not be better than 10 arcsec or calibration accuracies better than 2 percent
- (3) Background illumination is generally high (approximately 1000 ft-L) and non-uniform (clouds, reflections from structures, etc.)

The greatest drawback in the use of natural sunlight as the illumination source lies in the lack of control over atmospheric effects. Good seeing conditions are relatively rare, and under the most favorable conditions a noise equivalent angle of about 3 arcsec can be expected. Where high accuracy is important, consideration should be given to locating the test facility at as high an altitude as practicable, and away from objects that would tend to introduce nonreproducible thermal disturbances into the surrounding atmosphere (e.g., large parking lots, bodies of water, etc.). Much of the literature written for the astronomical community concerning seeing conditions is applicable to use of sunlight for sensor testing.

A useful technique for extending the limit for absolute accuracy is to image the Sun onto an aperture and then project this aperture to infinity via a second collimating system. The test area is located in the secondary beam, which, although fainter than direct sunlight, is less sensitive to atmospheric effects.

4.5.1.2 Artificial Sources

In general, no completely satisfactory solution to the problem of obtaining a good artificial illumination source exists today; however, the following guidelines are recommended for selecting sources best suited to various kinds of sun sensors:

- (1) For those cases where the brightness of the source is the most critical parameter and where uniformity and temporal stability are not paramount, the carbon or mercury-xenon sources are satisfactory. The mercury-xenon source has the advantage of cleanliness; however, spectral content is not optimal.
- (2) For those cases where the uniformity and temporal properties of the source are the most critical parameters, the tungsten blackbody cavities and hypereutectic graphite sources are superior. These sources have only a small fraction of the brightness of the Sun, but a successful test program can be carried out if one of the following design practices is adopted early in the program:
 - (a) The sun sensor and its electronics are designed to provide an adequate noise equivalent angle and adequate resolution at the input radiation level expected in testing
 - (b) Provision is made for an auxiliary amplifier of proper characteristics to boost the sun sensor signal to the space Sun levels
 - (c) The sensor system is designed to operate under the test source conditions and a neutral density filter is used to reduce the space Sun illumination level to the testing level—caution must be exercised to avoid introducing spurious reflection from the filter

In a typical installation, the primary source is imaged onto an aperture located at the focal point of a parabolic mirror. The mirror collimates the beam for transmission to the test area. Uniformity of the beam can be improved by the insertion of a diffuser (e.g., ground silica) at the aperture. The aperture of the simulator must be kept clean because contamination may cause a null shift. The “plunging” technique described in section 4.3.5 will detect a shift in the null axis.

4.5.1.3 Illumination in Environment Tests

Difficulties may be experienced in introducing the collimated beam into an environmental chamber, particularly when testing at low temperatures. Because of large thermal gradients across the chamber window, frost, stresses, and variations in index of refraction may result.

Frost problems may be reduced or eliminated by the use of dry nitrogen or helium gas in the chamber. Multiple layers of glass should be avoided. To avoid refraction problems, the chamber window should be maintained stationary with respect to the light beam as the sensor platform is rotated.

4.5.2 Calibration

The output characteristics of the sensor must be measured and extrapolated to conditions prevailing in space flight. Detailed calibration data are usually obtained for the sensor alone, i.e., prior to integration into the spacecraft system. Where possible, confirmation of preselected points or portions of the calibration curve should be checked after the sensor has been mated to the spacecraft (this is in addition to alignment) as a precaution against changes in calibration resulting from surrounding spacecraft surfaces.

Several points on the transfer function curve should be selected to serve as a baseline for detecting any anomalous behavior and changes in the transfer function that may occur during qualification and acceptance testing. Comparison of transfer functions measured before and after testing can reveal incipient failures. The use of this procedure exposed a problem associated with flaking of the anodized inner surface of the sensor casing on an early OSO spacecraft.

The preferred practice is to plot a continuous graph of the transfer function for each sensor for two orthogonal axes of rotation. This process can often be automated to reduce costs and the risk of human error.

Calibration data should be extrapolated to orbital conditions by either an analytical or an empirical procedure. In the analytical procedure, the sensor spectral responsivity is convolved with the test source radiance and the solar radiance to arrive at a correction factor. With care and an accurate knowledge of the spectral characteristics of sensor, illumination source, and the Sun, this procedure can yield results with accuracies in the range of 2 to 5 percent. Where higher accuracy correction factors are necessary, the empirical approach is recommended. The most common technique consists of comparing the sensor output under the test conditions with a standard cell whose output in space is known. A typical extrapolation procedure, including the effort expended in developing a standard, is discussed in reference 12.

APPENDIX

SELECTED TERMS IN SUN SENSOR TECHNOLOGY

Stability—the change in sensor output signal over a specified time and for a specified operating environment when a fixed angular relationship between the Sun and sensor coordinate system is maintained.

Absolute accuracy—the stability of the sensor for a time period beginning at sensor calibration and extending throughout the useful life of the sensor and for any combination of environments within the environmental specifications.

Absolute alinement accuracy—the angular stability of the sensor system relative to the spacecraft coordinates for a time period beginning at sensor calibration and extending throughout the useful life of the sensor and for any combination of environments within the environmental specifications.

Angular resolution—the smallest angular change consistently detected under the actual operating conditions or an accurate simulation of these conditions.

Noise equivalent angle—that angle through which the sensor must be rotated to generate a signal equivalent to the rms noise on the signal under the actual operating conditions or an accurate simulation thereof.

Null axis—that direction with respect to the reference coordinate system about which the sensor can be rotated without changing the sensor signal from a null or a reference zero value.

Reference coordinate system—the primary reference to which all sensor performance requirements are specified.

Transfer function—the relationship between the solar position within the sensor's coordinate system and the sensor's electrical output. Note: For analog sensors, the transfer function may conveniently be defined by a graph. For the digital sensor, a table, graph, or mathematical relationship may be used.

REFERENCES

1. Wilson, J. T.: Solar Radiation. Elec. Eng., vol. 66, no. 12, Dec. 1947, pp. 1209-1213.
2. Sharp, L.: History of Auto Sextants. Report, Kollsman Instrument Corp., no date.
3. Laszlo, T. S.; deDufour, W. F.; and Erdell, J.: A Guiding System for Solar Furnaces. Sol. Energy, vol. 2, no. 1, 1958, pp. 18-20.
4. Nidey, R. A.: Aspect Sensing, Astrostat Design and Orientation Control in Space Research. Preprint paper ARS 1016-59, ARS 14th Annual Meeting, Nov. 1959.
5. Bancroft, J. R.; Fain, M. Z.; and Johnson, D. F.: Final Report on the Design, Development, and Fabrication of a Solar Experiment Alignment Sensor (SEAS). NASA CR-73349, 1970.
6. Melugin, R. K.; Bancroft, J. R.; and Fain, M. Z.: A Solar Experiment Alignment Sensor. Proceedings of the Electro-Optical Systems Design Conference, Sept. 1970.
7. Melugin, R. K.; Johnson, D. F.; and Fain, M. Z.: Solar Eclipse Sensor. AIAA 2d Sounding Rocket Vehicle Technology Specialists Conference, Dec. 1970.
8. Bancroft, J. R.: Sun Sensor Design and Environmental Requirements Study for the Advanced Pioneer Spacecraft. Final Report. NASA CR-73086, Jan. 1967.
9. Massey, W. A.: Pioneer 6 Orientation Control System Design Survey. NASA/ERC Design Criteria Program, Guidance and Control. NASA CR-86071, 1968.
10. Harris, M.: Report on BBRC Radiation Resistant Sensors (S-17 MOD III Sensors). Tech. Note 64-22, Control System Design Section, Ball Brothers Research Corp., May 1964.
11. Staff of Lunar Orbiter Design Survey Team: Lunar Orbiter Program Guidance and Control System Design Survey. Sec. 8.2, Report D2-114277-3, Aerospace Systems Div., The Boeing Co., May 1969.
12. Fain, D. L.; Hall, J. M.; and Johnson, D. F.: Final Report on the Design, Development and Fabrication of a Precision Autocollimating Solar Sensor (PASS). Tech. Rept. IC033, Instrumentation and Control Div., Exotech, Inc., NASA CR-73260, June 1968.
13. Melugin, R. K.; Hall, J. M.; and Johnson, D. F.: A Precision Autocollimating Solar Sensor. AIAA Paper No. 69-858, Aug. 1969.
14. Staff, George C. Marshall Space Flight Center: The Meteoroid Satellite Project Pegasus. First Summary Report. NASA TN D-3505, 1966, pp. 61-62.
15. Anon.: Final Report Solar Pointing Aerobee Rocket Control System [SPARCS]. Sec. 2.1 of a Missiles & Space Co. report, Lockheed Aircraft Corp., 1968.

16. Anon.: Solar Electromagnetic Radiation. NASA SP-8005, June 1965.
17. Thekaekara, M. P.: Survey of the Literature on the Solar Constant and the Spectral Distribution of Solar Radiant Flux. NASA SP-74, 1965.
18. Stair, R.; and Ellis, H. T.: The Solar Constant Based on New Spectral II Radiance Data from 3100 to 5300 Angstroms. J. Appl. Meteorol., Aug. 1968.
19. Zirin, H.: The Solar Atmosphere. Blaisdell Pub. Co., 1965.
20. Tandberg-Hanssen, E.: Solar Activity. Blaisdell Pub. Co., 1967.
21. Fontana, A.: The Effect of Planetary Albedo on Solar Orientation of Spacecraft. NASA TN D-4133, Sept. 1967.
22. de Vaucouleurs, G.: Geometric and Photometric Parameters of the Terrestrial Planets. Memorandum RM-4000-NASA, Rand Corp., March 1964.
23. Winston, J. S.; and Taylor, V. R.: Atlas of World Maps of Long-Wave Radiation and Albedo for Seasons and Months Based on Measurements from Tiros IV and Tiros VII. Report NESC 43, National Environmental Satellite Center, Sept. 1967.
24. Sparkman, B. A.; and Cherrix, G. T.: Simultaneous Cloud Albedo Measurements Taken with Airborne Sol-A-Meters and Airborne and Nimbus II Orbiting Medium Resolution Infrared Radiometer. NASA TM X-55705, 1967.
25. Hutchings, P. B.: OGO Sun Sensor Assembly. Technical Description. Tech. Report 23-13-6019-TU-000 (NAS 5-899), TRW Systems, Oct. 23, 1963.
26. Anon.: Spacecraft Star Trackers. NASA SP-8026, July 1970.
27. Waitz, H.: Balloon-Borne Solar Pointing Control, Model 101. Tech. Manual TM 63-12, Ball Brothers Research Corp., Sept. 6, 1963.
28. Campen, C. F., Jr., ed.: Handbook of Geophysics for Air Force Designers. First ed., ch. 16, USAF-Cambridge Research Center, 1957.
29. Barrett, M. J.; and Stroud, R. H.: Proton-Induced Damage to Silicon Solar Cell Assemblies. A State-of-the-Art Survey. First Quarterly Report (JPL Contract 952246), Systems Research Div., Exotech, Inc., July 1968.
30. Jaffe, L. D.; and Rittenhouse, J. B.: Behavior of Materials in Space Environments. ARS Paper 2033-61, ARS Space Flight Report to the Nation (New York), Oct. 1961.
31. Anon.: Meteoroid Environment Model—1969 (Near Earth to Lunar Surface). NASA SP-8013, March 1969.
32. Anon.: Meteoroid Environment Model—1970 (Interplanetary and Planetary). NASA SP-8038, June 1970

NASA SPACE VEHICLE DESIGN CRITERIA MONOGRAPHS ISSUED TO DATE

SP-8001 (Structures)	Buffeting During Launch and Exit, May 1964
SP-8002 (Structures)	Flight-Loads Measurements During Launch and Exit, December 1964
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